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PRELIMINARY INVESTIGATION OF THE EFFECT OF COMPRESSIBILITY ON THE MAXIMUM LIFT COEFFICIENT

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SUMMARY

Preliminary data are presented on the variation of the maximum lift coefficient with Mach number. The data were obtained from tests in the 8-foot high-speed tunnel of three NACA 16-series airfoils of 1-foot chord. Measurements consisted primarily of pressure-distribution measurements in order to illustrate the nature of the phenomena.

It was found that the maximum lift coefficient of airfoils is markedly affected by compressibility even at Mach numbers as low as 0.2. At high Mach numbers pronounced increase of the maximum lift coefficient was found. The magnitude of the effects of compressibility on the maximum lift coefficient and the low speeds at which these effects first appear indicate clearly that consideration of the take-off thrust for propellers will give results seriously in error if these considerations are based on the usual low-speed maximum-lift-coefficient data generally used.

INTRODUCTION

The relative merits of different airfoil sections suitable for propellers are frequently evaluated by consideration of their critical speeds at low lift coefficient and their values of maximum lift coefficient as determined from low-speed tests. The basis of such a consideration is the assumption often made that because the speed is low in the take-off condition the compressibility effects may be ignored. Actually, compressibility effects may be of vital importance in the take-off condition because the propeller sections are operating at high lift coefficients for which the induced velocities are high and, consequently,

the critical speeds are very low. Because thrust available for take-off is generally determined by the magnitude of the maximum lift coefficient, it is therefore important to know the variation of the maximum lift coefficient with Mach number.

It has been shown in reference 1 and elsewhere that very large changes in the characteristics of an airfoil occur as the free-stream speed is increased to fairly large values. It has also been shown that the speed at which detrimental changes in the airfoil characteristics occur is a function of the lift coefficient and decreases with increase in lift coefficient. Calculation of critical speeds in the region of the maximum lift coefficient gives very low values. For ordinary airfoils, critical speeds of the order of 150 miles per hour at sea level are obtained. This low critical speed is, of course, a consequence of the very high local velocities over the leading edge portion of an airfoil at high angles of attack. Although shock may not form at the calculated value of the critical speed because of the critical form of the pressure-distribution diagram, separation is known to occur. Because the maximum lift phenomenon is essentially a separation phenomenon, large changes in the maximum lift coefficient might be expected. Detrimental effects have been found for an NACA 0012 airfoil at a Mach number as low as 0.2 (reference 2).

Take-off operation of propellers generally leads to tip Mach numbers as high as 0.9. Hence, even the inboard sections may show important compressibility effects on the maximum lift coefficient. Therefore, the whole take-off performance may be affected. With such operating conditions, a comparison of sections based on low-speed data is completely invalid.

Some data on the variation of maximum lift coefficient with Mach number are already available in references 1 and 3 and these data indicate that high values of maximum lift coefficient may be obtained in some instances. These data, however, are rather limited and no clear indication of the phenomena is indicated. The Reynolds number for these data is lower than the operating Reynolds number for propeller blade sections and this factor is important because propeller sections ordinarily operate in a critical Reynolds number range. A further limitation is due to tunnel-wall effects because of the relatively large size of model to tunnel, the ratio of the airfoil chord to the diameter of the tunnel being approximately 0.18.

Because of the importance of the effect of compressibility on the maximum lift coefficient and the incomplete data available, a detailed investigation of the phenomena has been planned by the NACA. The general investigation will include tests of several airfoils covering representative thickness and camber ranges. A part of the investigation has been completed and a part of the data so far obtained is reported herein.

The tests were conducted in the NACA 8-foot high-speed tunnel on models of 1-foot chord in order to obtain higher Reynolds number and reduced wall effects. The tests consisted, principally, of pressure-distribution measurements at one station at the center of the airfoil model, which spanned the tunnel. The results are essentially two-dimensional. Particular emphasis was placed on pressure-distribution tests rather than on force tests because the type of phenomena that occur is more clearly illustrated. The Mach number range extended from 0.12 to 0.53.

APPARATUS AND METHODS

The NACA 8-foot high-speed tunnel in which the tests were carried out is a single-return, circular-section, closed-throat tunnel. The airspeed is continuously controllable from about 75 to 550 miles per hour. The turbulence of the air stream as indicated by transition measurements on airfoils is unusually low but somewhat higher than in free air.

Three models having NACA 16-209, 16-509, and 16-515 airfoil sections of 1-foot chord were investigated. Thirty pressure orifices distributed along the chord were located at essentially the spanwise station at the center of the air stream. The orifice locations and airfoil shapes are shown in figure 1. The airfoil ordinates were calculated by the methods described in reference 4 and are given in table I.

The model, when mounted in the tunnel, completely spanned the jet (fig. 2). Except for auxiliary streamline wire bracing, required because of structural considerations, the standard 8-foot high-speed tunnel model mounting and setup were employed. Tests at low and medium speeds with and without the braces indicated that interference of the

auxiliary supports on the flow at the measurement station was negligible.

Measurements consisted, principally, of the chordwise pressure distribution at the midspan region. The surface orifices in the airfoil were connected to a multiple-tube manometer located outside the test section. The pressure tubing connecting the orifices was of small diameter and was located within the wing. Simultaneous recording of the pressures at all orifices in the wing was made by photographing the multiple-tube manometer.

The Mach number range extended from 0.12 to 0.53. The Reynolds number range of the tests and the variation of the Reynolds number with the Mach number are shown in figure 3. There is also included in figure 3 a curve showing the relation between the Reynolds number and the Mach number for unpublished tests of an NACA 0012 airfoil made in the 19-foot pressure tunnel, data for which are included in this report for purpose of comparison.

RESULTS

The maximum lift coefficient $C_{L_{max}}$ was determined as the highest value obtained in the positive angle-of-attack range. The angle of attack at which maximum lift occurred varied with the Mach number M. The maximum lift for the 16-209 airfoil was found to occur at angles of attack between 8° and 10° with no consistent trend with Mach number. For the 16-509 airfoil, the maximum lift occurred at 10° at low Mach number and decreased with increase of Mach number to 8° at a Mach number of 0.53. For the 16-515 airfoil, the angle of attack for maximum lift was 15° for Mach numbers up to 0.25 and decreased to 12° at a Mach number of 0.53. The values of the maximum lift coefficient as presented in this report are approximate values, having been determined as the product of the integrated normal-force coefficient and the cosine of the angle of attack. The drag component has not been included. Neglect of the drag component generally involves small effects and does not influence the character of the variation of the maximum lift coefficient with Mach number to any great degree.

The variation of the maximum lift coefficient with Mach number is given in figure 4. Pressure-distribution diagrams for angles of attack in the maximum-lift-coefficient range are given in figures 5 to 8. In these figures, the pressure

coefficient P is defined as $\Delta p/q$ where Δp is the difference between local static pressure and free-stream static pressure and q is the dynamic pressure of the free stream. For the higher speeds, only partial pressure distributions are shown. Outside of the regions shown, the results are the same as for the lower speed.

There is also included in figure 4 the variation of the maximum lift with Mach number for an NACA 0012 airfoil as found from tests in the NACA 19-foot pressure tunnel. These data, which were taken from reference 2, were from force tests of a wing having an aspect ratio of 6 and the tip effects are therefore included; whereas the data obtained in the present investigation are essentially two-dimensional. Other unpublished data indicate the effect of the tips may slightly increase the critical Mach number. It is not believed likely, however, that the character of the variation of maximum lift with Mach number will be seriously altered by the tip effect.

DISCUSSION

Of the factors affecting variation of the maximum lift coefficient, the effects of Reynolds number are generally understood except when an increase of Reynolds number is accompanied by an increase of Mach number. With increase of Mach number, compressibility effects become pronounced and, in general, lead to separation phenomena with consequent decrease in the value of the maximum lift coefficient. As has previously been shown, compressibility effects on the maximum lift coefficient may predominate even at a Mach number as low as 0.2 (reference 2). For data obtained in this investigation both the Reynolds number and the Mach number vary. At very low speeds the Reynolds number effects predominate but at higher speeds the compressibility effects predominate. The camber and the thickness ratio, particularly as the nose shape may be affected, can also have a large effect on the maximum lift coefficient and its variation with either Reynolds number or Mach number.

The variation of the maximum lift coefficient plotted against the Mach number is shown in figure 4 for the three airfoil sections, NACA 16-209, 16-509, and 16-515. The character of the variation of the maximum lift coefficient is indicated to be a function of the camber and the thick-

ness ratio. For the NACA 16-209 airfoil the maximum lift coefficient is essentially constant. This result is in accord with previous results which have demonstrated that thin airfoils, because they generally have small leading-edge radii, have essentially fixed separation points. An increase of camber markedly alters both the value of the maximum lift coefficient and the variation with Mach number, as is shown by the results for the NACA 16-509 airfoil. For this airfoil the maximum lift coefficient increases until a Mach number of approximately 0.25 is obtained. Between Mach numbers of 0.25 and 0.40 the maximum lift coefficient is essentially constant but increases markedly as the Mach number exceeds 0.40. An increase of thickness increases to a very great extent the magnitude of the general effects observed for the thinner 16-509 airfoil as is indicated by the data for the 16-515 airfoil.

The increase in the maximum lift coefficient observed at low speeds is similar to the Reynolds number effect commonly observed for airfoils of medium thickness. An increase of the Mach number, however, leads to larger adverse pressure gradients back of the peak pressure points and, hence, to the tendency toward earlier separation. With continued increase in Mach number, the separation tendency becomes greater until finally any favorable effect due to increase in Reynolds number is so counteracted that there is either no increase or, as in the case of the NACA 16-515 airfoil, a fairly large decrease in the maximum lift coefficient. The fundamental mechanism of these phenomena is not clearly understood and considerable further investigation is necessary.

The large increase in the maximum lift coefficient at high Mach numbers is due to rearward movement of the peak negative pressure, as is illustrated by the data in figures 5, 6, and 7. At a Mach number of 0.40 the shape of the pressure distribution diagram is typical for the maximum lift coefficient region at low speeds where a high negative pressure peak occurs slightly aft the leading edge. With increase of speed compression shock occurs, which reduces the pressure peak at the nose, and with further increase in speed the shock moves rearward giving rise to more extensive regions of high negative pressure. The loss of lift resulting from the decrease of the local peak that occurs at the nose at low speed is then offset by the more extensive low-pressure region. The effects of compressibility are similar to those found at lower lift coefficients in regard to the formation and rearward movement of shock and the location of the peak negative pressure (reference 5).

The failure of the NACA 16-209 airfoil to show this increase in maximum lift coefficient cannot be interpreted to mean that a different phenomenon occurs. An examination of the pressure-distribution data presented in figure 8 for this airfoil shows that the maximum negative pressure coefficient is slightly less than -2.0. The critical Mach number corresponding to this pressure coefficient is slightly above the highest Mach number reached in these tests and, hence, the shock phenomenon leading to the increase of the maximum lift coefficient is not encountered. At higher speeds where shock phenomena would be encountered, the maximum lift coefficient for this airfoil should be expected to show the sharp rise. The need for further investigation of thin low-cambered airfoils is indicated.

The increase in the value of the maximum lift coefficient of approximately 0.25 with increase of airfoil design lift coefficient (camber) as shown by the data for the 16-209 and 16-509 airfoils in figure 4 is noteworthy. This effect is important, practically, in relation to the take-off problem for propellers. The blade sections may be designed with slightly higher camber than would be dictated by the design high-speed lift coefficient to obtain improvement in take-off. This improvement can be effected without encountering any serious compressibility loss at high speeds because this class of airfoil has a range of lift coefficients about the design value for which the critical Mach number is sensibly constant. Thus, through the use of slight overcamber, a higher blade-section maximum lift coefficient can be obtained with the consequent improvement in the thrust for the take-off condition without materially affecting the high-speed operation.

Comparison of data for an NACA 0012 airfoil taken from unpublished tests made in the 19-foot pressure tunnel shows the same general variation of lift coefficient as do the NACA 16-515 and the 16-509 airfoils in the low and intermediate speed range. The Reynolds number effect is counteracted by the compressibility effects at a Mach number of approximately 0.20. The higher values of maximum lift coefficient in the low-speed range are due probably to the higher Reynolds number (fig. 3) at which the tests of this airfoil were made. The large variation encountered may indicate more critical compressibility effects for this type of airfoil section.

SUMMARY OF RESULTS

1. The maximum lift coefficient of airfoils is affected by compressibility to a marked degree at Mach numbers as low as 0.2.

2. At high Mach numbers pronounced increases in maximum lift coefficient occur.

3. The low speed for which compressibility effects on the maximum lift coefficient may be encountered indicates that the maximum thrust available for take-off may be markedly affected. Selection of sections suitable for propellers must, therefore, be made with due consideration of the effect of compressibility on the maximum lift coefficient.

4. It is indicated that, for marginal take-off thrust, some improvement may be effected without serious loss at high speed by use of design camber slightly higher than would be dictated by high-speed considerations alone.

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2. Stack, John: Compressibility Effects in Aeronautical Engineering. NACA A.C.R., Aug. 1941.

3. Stack, John: The N.A.C.A. High-Speed Wind Tunnel and Tests of Six Propeller Sections. Rep. No. 463, NACA, 1933.

4. Stack, John: Tests of Airfoils Designed to Delay the Compressibility Burble. NACA A.C.R., June 1939.

5. Stack, John, Lindsey, W. F., and Littell, Robert E.: The Compressibility Burble and the Effect of Compressibility on Pressures and Forces Acting on an Airfoil. Rep. No. 646, NACA, 1938.

TABLE I

AIRFOIL ORDINATES

NACA 16-209			
UPPER SURFACE		LOWER SURFACE	
x (percent)	y (percent)	x (percent)	y (percent)
0.256	0.512	0.344	-0.448
.545	.733	.655	-.615
1.183	1.074	1.317	-.860
2.421	1.538	2.579	-1.166
4.912	2.196	5.088	-1.564
7.409	2.696	7.591	-1.848
9.909	3.109	10.091	-2.075
14.914	3.772	15.086	-2.426
19.923	4.293	20.077	-2.701
24.933	4.708	25.067	-2.918
29.945	5.035	30.055	-3.091
34.958	5.285	35.042	-3.225
39.972	5.461	40.028	-3.319
44.986	5.568	45.014	-3.378
50.000	5.603	50.000	-3.397
55.014	5.566	54.986	-3.376
60.028	5.447	59.972	-3.305
65.041	5.237	64.959	-3.177
70.053	4.924	69.974	-2.980
75.063	4.497	74.937	-2.707
80.069	3.943	79.931	-2.351
85.071	3.253	84.929	-1.909
90.066	2.404	89.934	-1.370
92.061	2.017	91.939	-1.129
94.054	1.598	93.946	-.876
95.050	1.375	94.950	-.743
96.049	1.145	95.956	-.611
98.031	.651	97.969	-.339
100.000	.089	99.989	-.089

Slope of radius
through end of chord = 0.124
L.E. radius = 0.396 percent

NACA 16-509			
UPPER SURFACE		LOWER SURFACE	
x (percent)	y (percent)	x (percent)	y (percent)
0.192	0.550	0.408	-0.390
.465	.810	.735	-.514
1.084	1.223	1.416	-.687
2.305	1.805	2.695	-.875
4.781	2.659	5.219	-1.079
7.274	3.323	7.726	-1.203
9.774	3.876	10.226	-1.292
14.787	4.775	15.213	-1.411
19.807	5.483	20.193	-1.503
24.833	6.048	25.167	-1.572
29.863	6.491	30.137	-1.631
34.895	6.829	35.105	-1.679
39.929	7.067	40.071	-1.711
44.964	7.211	45.036	-1.735
50.000	7.258	50.000	-1.742
55.036	7.209	54.964	-1.733
60.071	7.053	59.929	-1.697
65.104	6.781	64.896	-1.631
70.133	6.380	69.867	-1.520
75.157	5.838	74.843	-1.362
80.173	5.133	79.827	-1.153
85.178	4.257	84.822	-.893
90.164	3.173	89.836	-.589
92.152	2.677	91.848	-.457
94.135	2.133	93.865	-.329
95.123	1.843	94.877	-.263
96.110	1.540	95.890	-.204
98.076	.880	97.924	-.100
100.000	.000	100.000	.000

Slope of radius
through end of chord = 0.312
L.E. radius = 0.396 percent

NACA 16-515			
UPPER SURFACE		LOWER SURFACE	
x (percent)	y (percent)	x (percent)	y (percent)
0.120	0.861	0.480	-0.701
.375	1.253	.825	-.957
.973	1.859	1.527	-1.323
2.174	2.699	2.826	-1.769
4.635	3.907	5.365	-2.327
7.123	4.831	7.877	-2.711
9.624	5.598	10.376	-3.014
14.644	6.838	15.356	-3.474
19.679	7.811	20.321	-3.831
24.722	8.588	25.278	-4.112
29.772	9.197	30.228	-4.337
34.825	9.664	35.175	-4.514
39.882	9.995	40.118	-4.639
44.941	10.193	45.059	-4.717
50.000	10.258	50.000	-4.742
55.059	10.189	54.941	-4.713
60.118	9.969	59.882	-4.613
65.173	9.584	64.827	-4.434
70.222	9.014	69.778	-4.154
75.262	8.237	74.738	-3.761
80.289	7.230	79.711	-3.250
85.296	5.973	84.704	-2.609
90.274	4.428	89.726	-1.844
92.254	3.720	91.746	-1.500
94.224	2.952	93.776	-1.148
95.206	2.546	94.794	-.966
96.184	2.121	95.816	-.785
98.127	1.208	97.873	-.428
100.044	.045	99.956	-.045

Slope of radius
through end of chord = 0.312
L.E. radius = 1.10 percent

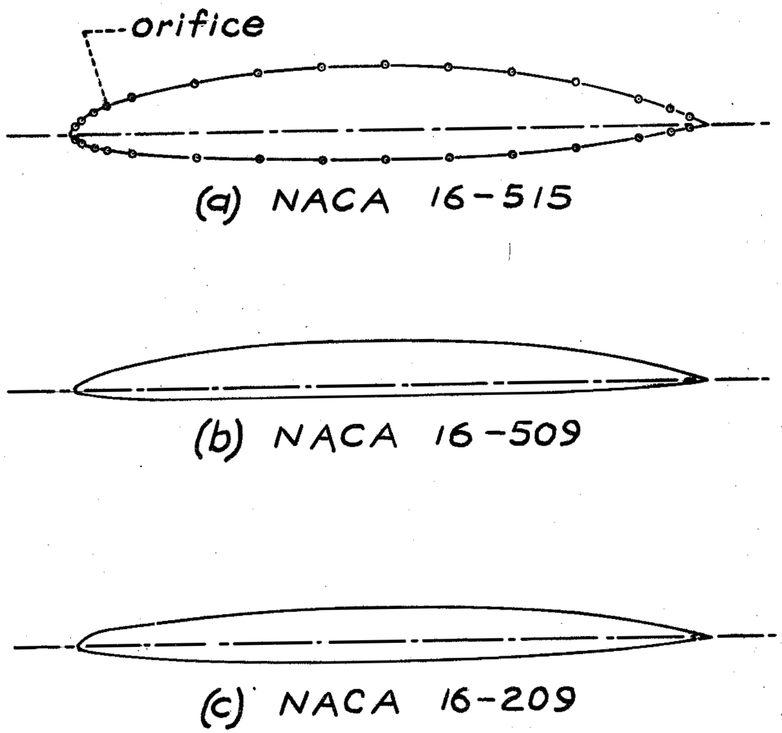

Figure 1.– Airfoils tested. Orifice locations the same for all airfoils.

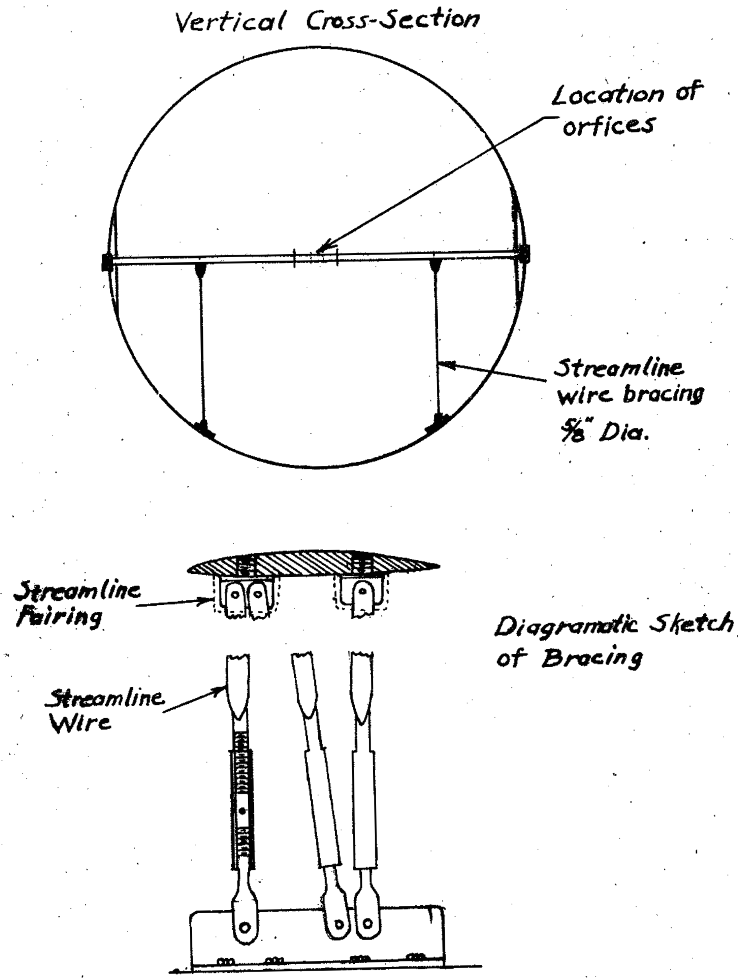

Figure 2.– Diagramatic sketch of wing mounted in 8-foot high-speed tunnel.

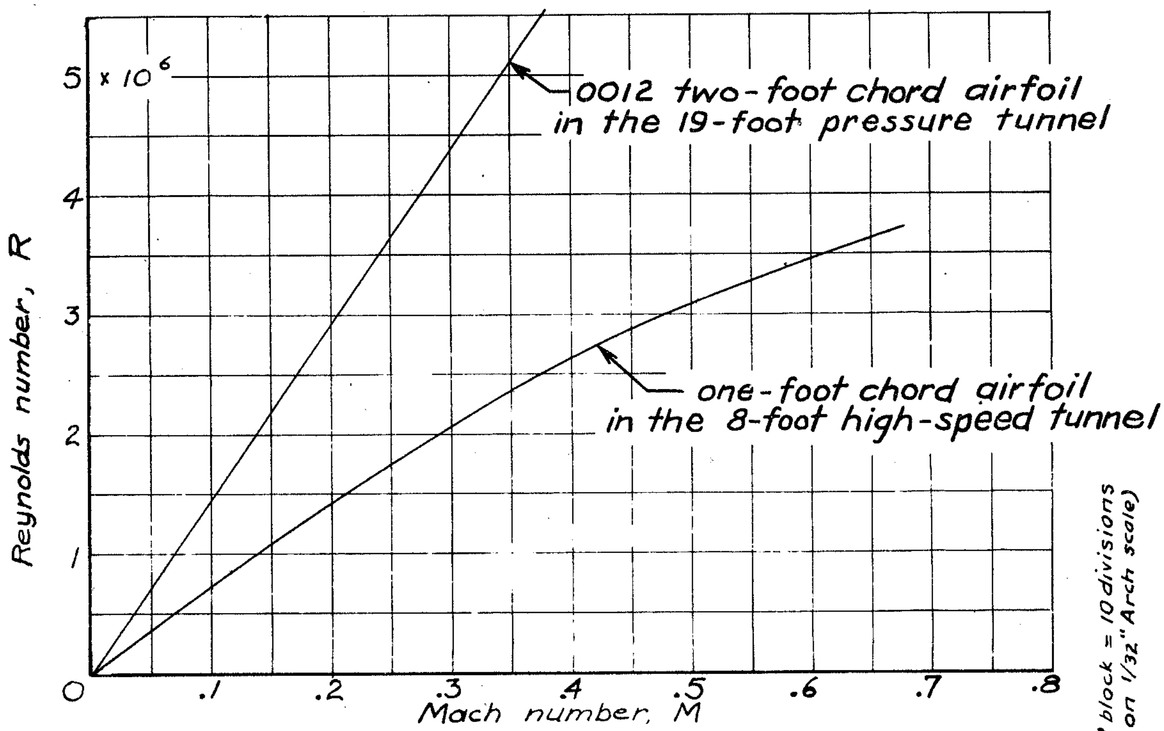

Figure 3.- Variation of Reynolds number with Mach number.

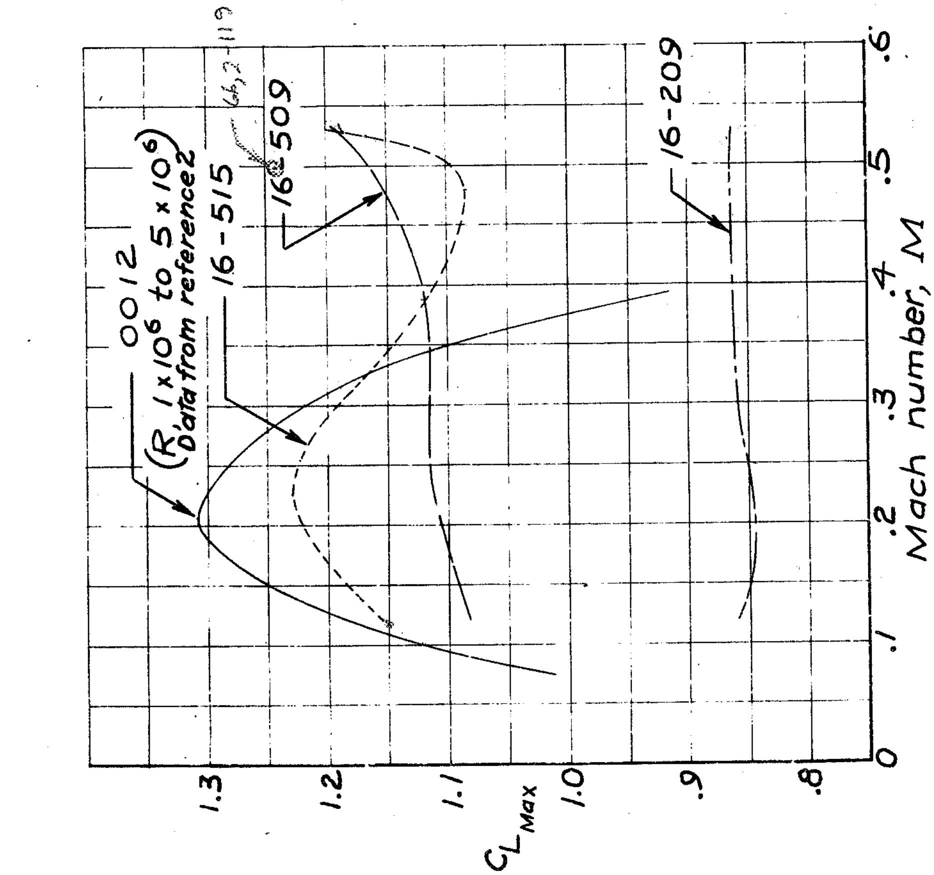

Figure 4.- Variation of maximum lift coefficient with Mach number for three one-foot chord 16-series airfoils, $R = 8.5 \times 10^5$ to 3.25×10^6, and for the NACA 0012 two-foot chord airfoil.

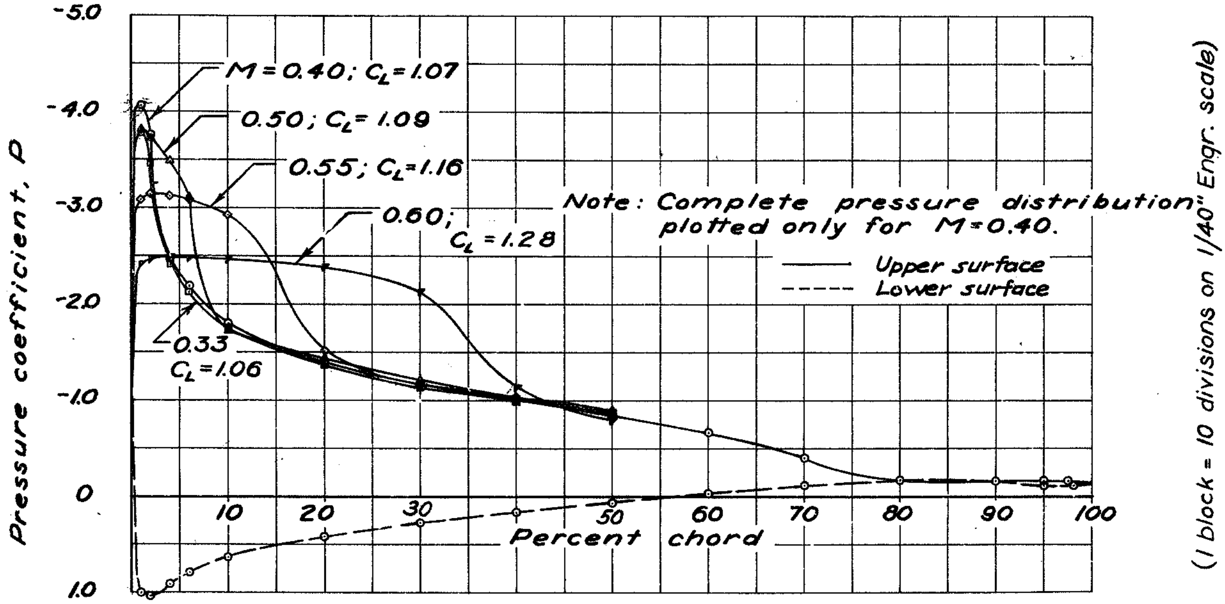

Figure 5.- Variation in pressure distribution over the forward 50 percent of the upper surface with Mach number for the NACA 16-515 at an angle of attack of 11 degrees.

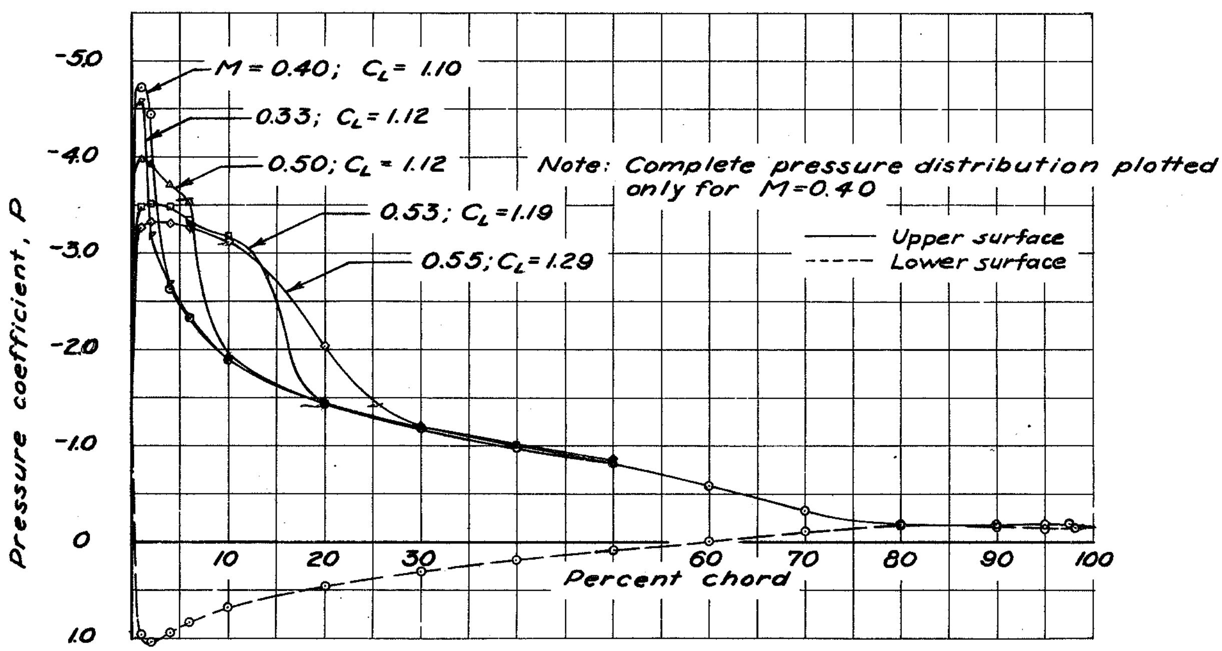

Figure 6.- Variation in pressure distribution over forward 50 percent of the upper surface with Mach number for the NACA 16-515 at an angle of attack of 12 degrees.

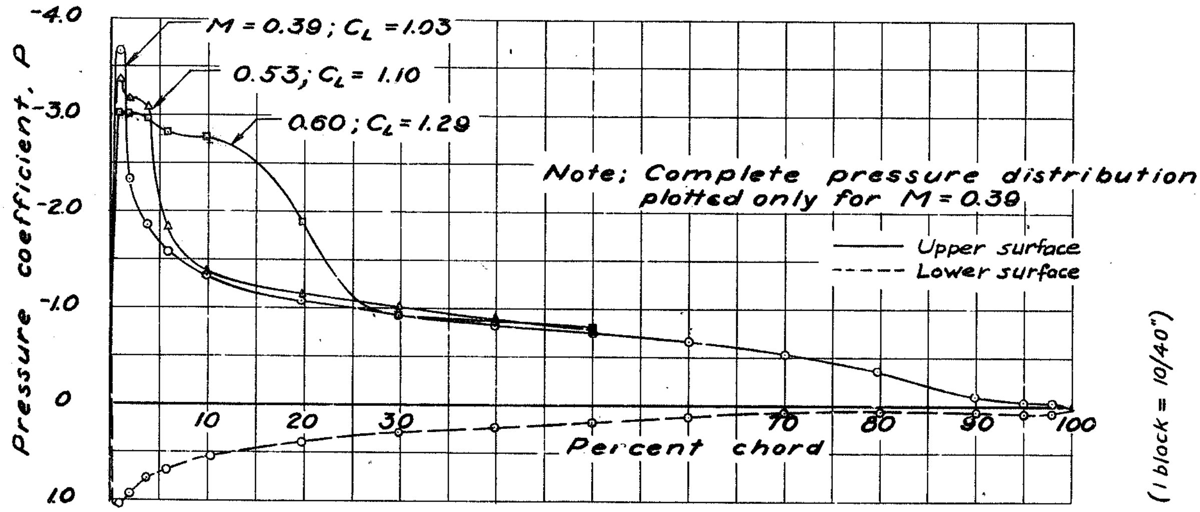

Figure 7.-Variation in pressure distribution over the forward 50 percent of the upper surface with Mach number for the NACA 16-509 at an angle of attack of 7 degrees.

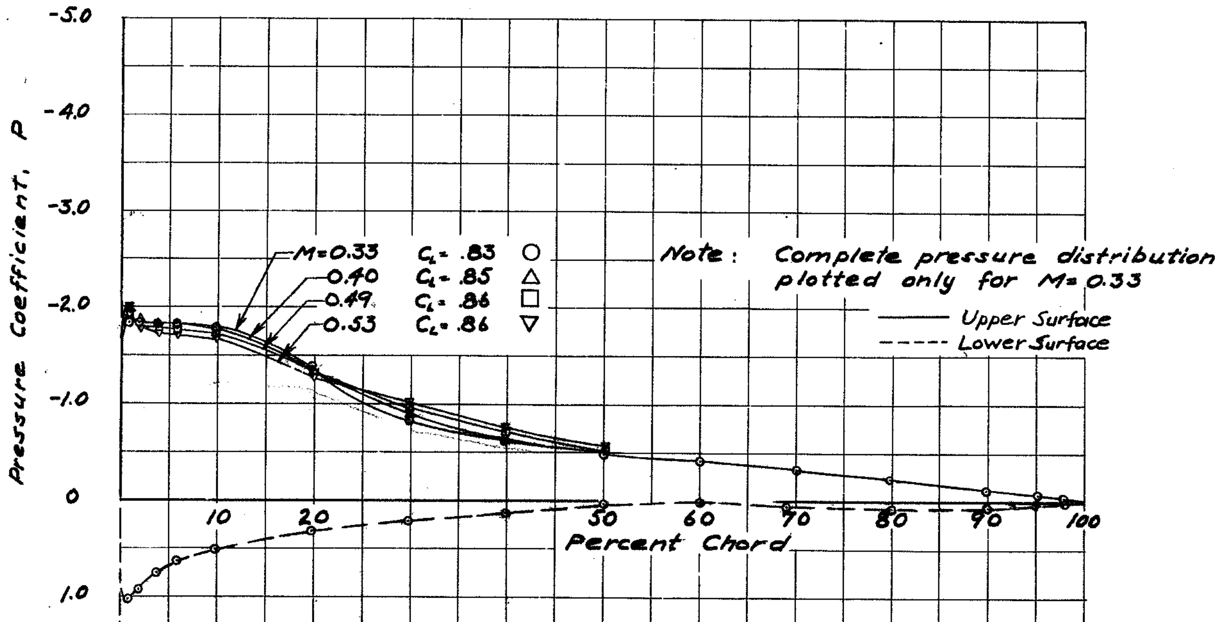

Figure 8.-
Variation in pressure distribution over the forward 50 percent of the upper surface with Mach number for the NACA 16-209 at an angle of attack of 8 degrees.